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JET VANE HEAT TRANSFER MODELING

by

Robert H. Nunn

and

Matthew D. Kelleher

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JET VANE HEAT TRANSFER MODELING

A Report Prepared for

Naval Weapons Center

China Lake, California

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Robert H. Nunn

and

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Naval Postgraduate School

Monterey, California

October 1, 1986

ABSTRACT

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JET VANE HEAT TRANSFER MODELLING

INTRODUCTION

Jet Vane TVC Systems

Thrust vector control (TVC) systems offer means of trajectory control that are virtually independent of external forces. Such capability is frequently required for tactical missiles, as well as spacecraft launch vehicles, when the relative flow past external lifting surfaces is insufficient to generate the necessary control forces. This commonly occurs during low-speed operations, such as at launch or during hovering flight. High angle of attack flight may also lead to regimes where conventional lifting surfaces are inadequate. In addition, there are occasions when external steering devices are infeasible from a design point of view, such as for tube-launched devices.

Several methods of TVC have been developed and applied to operational and experimental vehicles [1,2]. These include movable nozzles, internal fluid injection (secondary injection), and mechanical jet deflection systems. Jet vane systems fall in the latter category and they tend to be favored for volume-limited applications requiring relatively low actu-

ation torques and rapid response. Among the various methods of TVC, jet vanes have been found to be the most effective in delivering large thrust deflection angles (up to about 30^{o}). They may also be used with relative ease to generate roll torques.

The application of jet vane TVC dates back to the rockets designed by Goddard, and has extended to the Redstone, Sergeant, Talos, Pershing, and Algo II and III motors [1,2], as well as several intallations in smaller tactical rockets.

Of course, there are disadvantages accompanying the selection of jet vanes for TVC purposes. These include thrust losses on the order of 3-5% with undeflected vanes [2]. In addition, the attainment of relatively high thrust deflection angles may lead to axial thrust losses of the same order of magnitude as the resulting side force. However, the chief problem associated with the use of jet vanes is the large thermal loading that they experience as they are required to operate in hot, high-speed, particle-laden flows. This problem leads to design limitations so that jet vanes are often restricted to short-duration use in motors with low-temperature non-metalized propellants.

The aerodynamic (side-force producing) characteristics of jet vanes may be calculated with fair certainty on the basis of inviscid flow theory with suitable corrections for viscous effects [2,3]. On the other hand, difficulties that stem from the severity of the jet-vane thermal environment tend to lead to design practices that are based upon past experience and

cut-and-try methods. Over-design is therefore inevitable, with virtually no capacity for design optimization. In order to exploit the several advantages of jet vane TVC systems, therefore, it has become necessary to build reliable data bases and, to the extent possible, attain a fundamental understanding of the heat transfer aspects of such systems. This has led to the work reported herein.

The Jet Vane Thermofluid Environment

In the design of a jet vane system, the integrity of the vanes themselves must be guaranteed over the specified work cycle. Although this is a serious challenge, designers must also also consider the behavior of the vanes and supporting structure during transient events. Upon motor ignition, localized jet vane temperatures may rise from their initial values to near-stagnation temperatures in fractions of a second. Temperatures in the vane attachment device will rapidly follow this rise, and large thermal stresses may develop due to the proximity of a relatively cool supporting structure. These and other design aspects can only be addressed with precision if there is a good understanding of the convective heat transfer process that gives rise to the energy transfer from the flowing gases to the the vane. Put another way, the application of complex computer codes for thermal conduction in the vane and supporting structure can only follow the specification of the convective boundary conditions.

With respect to these conditions, the problem is even

more complex. The vane is immersed in a flow field that is, if generally described, compressible, turbulent, multi-component (and possibly multi-phase), three-dimensional, and unsteady, with variable properties and nonlinear and time-variant boundary conditions. These complexities, if taken more than one at a time, present a problem that is beyond the state-of-the-art for exact solution. In addition, the overall flow field will contain intersecting and impinging shock waves that give rise to discontinuous events and further complication of the boundary conditions. The presence of various protuberances only serves to exacerbate these difficulties.

To further define the problem, it may be of use to consider the levels of heat transfer that might be expected from the point of view of "simple" convection from a supersonic flow to a cooled wall. The convective heat transfer coefficient (h) may be described in terms of the nondimensional Stanton number (St) as follows:

$$h = (\ Vc_p)\ St$$

where , V, and c_p are the density, velocity, and specific heat at constant pressure of the flowing gas. On the assumption that the gas behaves ideally, this expression may be written

$$h = St\ \left(\frac{k}{k-1}\ \frac{P}{T}\ g\ I_{sp}\right)$$

where P and T are local gas pressure and temperature and I_{sp} is the specific impulse. Consider, for instance, the condi-

tions at the exit of a rocket nozzle flowing at Mach 3. With a ratio of specific heats assumed to be k = 1.2, we have, approximately:

$$h = St\ (0.25\ \frac{P_o}{T_o}\ g\ I_{sp})$$

where the subscript $(_o)$ refers to stagnation conditions. If a typical case is taken to be given by P_o = 100 bar, T_o = 2560K, and I_{sp} = 250 s, then, in round numbers,

$$h = 2000\ St\ (kW/m^2K)$$

In rocket engine nozzles, a typical value for the Stanton number is about 0.002 [4]. Thus, with the result above, the expected value of the heat transfer coefficient is on the order of 4 kW/m^2K. Such a value should not be taken as conservative, since actual motor conditions may be more severe and local regions (e.g. stagnation points) may experience much higher values. By way of comparison, values of h such as these are well beyond those usually considered for convective heat transfer and, in fact, they are more typical of those realized in phase-change processes [5].

Method of Approach

In short, there can be no argument that a complete and general analytical solution is too much to hope for. On the other hand, rapid advancements in the field of computational fluid mechanics give reason to expect that much progress can be made in predicting the major features of the flow field and

their effects on heat transfer to the vane. Even so, it will be necessary to provide considerable experimental backup in order to verify numerical predictions and to evaluate the effects of factors that are beyond present-day computational capabilities.

This report describes the initial efforts in such a combined computational-experimental program. In the next section a brief review is given of some of the existing knowledge that may be brought to bear in the design of jet vane TVC systems. This is followed by a description of the code PHOENICS and discussion of some of the preliminary results that have been obtained through its use. The progress of the experimental program then will be summarized, again with some preliminary results.

BACKGROUND

This section of the report has been prepared in order to provide an overview of some of the previous work that has been accomplished in the analysis of the heat transfer aspects of supersonic flows. The particular results that are discussed are meant to prove useful as "first-cut" design guidelines for the estimation of the thermal loads applicable to the jet vane configuration. With this goal in mind, an effort has been made to identify relationships that are backed by experimental results. Purely theoretical results will not receive attention here.

It is also intended that the reference list provided will serve as a starting point for those wishing to conduct a more-complete review of the available literature. Only the unclassified open literature is discussed here, with the result that some information of a practical nature may be omitted. On the other hand, it is felt that most proven results that are widely applicable will have found their way into the public domain. For those seeking information relative to particular applications, the Chemical Propulsion Information Agency has provided a useful bibliography [6].

In searching the literature (albeit in a rather cursory way), it has been observed that relevant articles seem to be grouped in time with large intervening gaps. A surge of information appears in the mid-50s, in apparent response to stimuli provided by Sputnik and the emergence of supersonic flight as an accepted mode of transportation. The overall

impact of this work was to provide what was then seen as an adequate basis for the prediction of supersonic effects such as the aerodynamic heating of simple configurations. The complexity of the general thermal problem -- including, for instance, shock wave interactions, dissociation at high speeds, and complicated geometries resulting in three-dimensional flows -- was only accounted-for by correction factors and overdesign. Since then, with a few noteworthy exceptions, the awesome nature of the complete problem has seemingly precluded major advances in the general understanding of heat transfer in supersonic flows.

Since the mid-50s, much work has transpired in the understanding of shock wave-boundary layer interactions. Secondly, the advent of space exploration has led to significant developments in the state-of-the-art associated with hypersonic flight (with great emphasis on the space shuttle). With respect to experimental investigations, these advances have been greatly enhanced by the increasing use of facilities such as shock tunnels, gun tunnels, and Ludwieg tubes [7]. The third major factor associated with high-speed flows has been the explosive growth of the field of computational fluid mechanics. In this area, researchers are now able to predict flow effects that are most difficult, and sometimes impossible to measure with precision. From the point of view of this brief summary, neither of these latter two areas of advancement are particularly relevant.

Insofar as thermal effects are concerned, there appears

to be a relative dearth of empirical design information that is directly and generally applicable to flows at supersonic Mach numbers below about six. For flows with intact boundary layers, the results of the mid-50s still seem to be applicable, with relatively few extensions since then. We hasten to repeat, however, that this conclusion is based on a view of the literature that is far from comprehensive.

In the following discussion, initial emphasis is placed upon the thermal transport across an intact boundary layer separating a stationary flat plate from a supersonic freestream flow. Following this, some departures from this more-or-less ideal situation are discussed, such as shock wave-boundary layer interactions and the effect of suface protuberances.

Heat Transfer in Supersonic Boundary Layer Flows.

The heat transfer to or from a bounding wall occurs on a microscopic scale at the wall. The rate of heat transfer is proportional to the temperature gradient at the wall, and increases or decreases according to variations in this gradient. For given temperatures of the wall and the external freestream (outside of the boundary layer), variations in the wall temperature gradient follow variations in the freestream velocity. Increased speeds lead to thinning of the boundary layer, with the temperature change between freestream and wall occuring across the decreased thickness -- temperature gradients and attendant heat transfer rates increase with free-

stream speed, and vice-versa.

In high speed flows, these effects may be altered significantly by direct compression or viscous dissipation of energy by friction within the boundary layer. In such instances, fluid temperatures within the boundary layer may exceed freestream temperatures with attendant increases in heat transfer to a wall that is maintained below the freestream temperature. In cases where the wall temperature is maintained above the freestream temperature, heat may be first transferred <u>from</u> the wall at low speeds and then <u>to</u> the wall as the flow speed is increased.

In high speed flows, therefore, there is a certain wall temperature, above freestream temperature, at which there is no heat transfer to the wall. This temperature is called the "recovery" or "adiabatic wall" temperature. Whereas in low speed flows the freestream temperature is the reference temperature for heat transfer, the proper reference temperature in high speed flows is the recovery temperature. The recovery temperature, T_r, is defined in terms of the recover factor, r, as follows:

$$T_r = T_1[1 + r(\frac{T_o}{T_1} - 1)]$$

where the subscript $(_1)$ is used to denote conditions at the boundary layer edge. Here, T_o is the maximum freestream temperature theoretically achievable (stagnation temperature):

$$T_o = T_1(1 + \frac{k-1}{2} M_1^2)$$

so that

$$T_r = T_1(1 + r \frac{k-1}{2} M_1^2)$$

The calculation of heat transfer in high speed flows thus requires a determination of the recovery factor. The second important factor that distinquishes high speed flows from their incompressible counterparts is the fact that fluid properties are, in general, dependent upon temperature. With the severity of temperature variations in such flows, and the possibility of variations in density as well, the use of appropriate fluid properties becomes a matter of great importance.

As has been mentioned, a large amount of relevant information has been reported in the mid-50s timeframe. This is amply documented in a number of excellent review articles, among them Kaye [8], Eckert [9], and van Driest [10,11]. Applications to rocket motors are discussed in Summerfield [12] and, more recently, Ziebland and Parkinson [4]. In the following sections, we attempt to present the main features of these works.

Laminar Boundary Layers. For incompressible laminar boundary layer flow, application of the Reynolds analogy leads to a Stanton number given by:

$$St = h/ c_p V = 0.33/(Re^{1/2} Pr^{2/3})$$

It has been found that the form of this result can be carried-over into compressible flows provided that the recovery temperature, as given above in terms of recovery factor and free-

stream Mach number and temperature, is used as the reference temperature for heat transfer. That is, the heat transfer rate from the wall, per-unit-area, q'', is given by

$$q'' = h(T_w - T_r)$$

In another form,

$$q'' = h(T_w - T_1)\ [1 - f_1(r, M_1, T_w/T_1)]$$

where

$$f_1(r, M_1, T_w/T_1) = r\ \left(\frac{\frac{k-1}{2} M_1^2}{\frac{T_w}{T_1} - 1}\right)$$

which illustrates that the modification for compressible flow depends upon both M_1 and T_w/T_1.

For laminar flow, the recovery factor, and hence the recovery temperature, may be calculated with 1% accuracy by

$$r = Pr^{1/2}$$

for Prandtl numbers in the range $0.7 < Pr < 1.2$ and Mach numbers up to about 8 [8].

Use of these expressions further requires that fluid properties be evaluated at specified conditions. References [8] and [11] recommend the use of the freestream temperature for this evaluation, leading to the results shown in Fig. 1 for air with a Prandtl number assumed constant at 0.75. Note that these results lead to a value of the "constant" coefficient for the Stanton number that depends upon the Mach number and the temperature ratio in compressible flow. The question of the correct reference temperature for the evaluation of fluid properties is further addressed by Eckert [9],

in which this temperature is given the symbol T^* and is defined by:

$$\frac{T^* - T_1}{T_w - T_1} = 0.5 + 0.22\, f_1(r, M_1, T_w/T_1)$$

where f_1 is defined above.

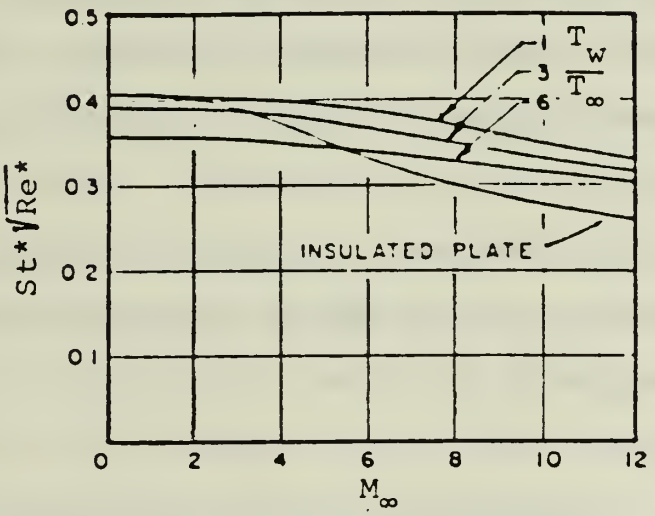

Figure 1. Local Stanton number for a flat plate in air, Pr = 0.75 [8].

In summary, for high speed flows with laminar boundary layers the Stanton number may be estimated as follows:

$$St^* = 0.33/(Re^{*1/2} Pr^{*2/3})$$

where the star notation indicates that fluid properties are to be calculated using T^*. From this, the film coefficient may be evaluated and hence the heat transfer rate per-unit-area, based upon the difference between freestream and recovery temperatures.

Turbulent Boundary Layers. The details of turbulent boundary layer flows are poorly understood -- today, as they

were in the mid-50s. Turbulent flow heat transfer formulations, based upon the Reynolds analogy between skin friction and heat transfer are, of course encumbered by the same uncertainties. With the acceptance of their approximate nature, the adaptation of incompressible formulas to compressible flows follows the same basic philosophy as for laminar flows. That is, the reference temperature for heat transfer is the recovery temperature and the fluid properties must be evaluated at T^*.

For turbulent boundary layers, the recovery factor likewise has been found to be mainly dependent upon the Prandtl number, as follows:

$$r = Pr^{1/3}$$

The resulting recommended expressions for Stanton number are

$$St^* = 0.0296/(Re^{*1/5} Pr^{*2/3}), \quad \text{for } 5\times10^5 < Re < 10^7$$

and

$$St^* = 0.185/[Pr^{*2/3} (\log_{10}Re^*)^{2.58}], \quad \text{for } Re > 10^7$$

These expressions assume a smooth plate surface. Modifications for rough surfaces are necessary, but extensive systematic information on rough surfaces exists only for the friction coefficient in low speed flows.

For both laminar and turbulent boundary layers, these expressions for high-speed flow are still recommended today for purposes of preliminary design [5].

Other_Expected_Features_in_Jet_Vane_Heat_Transfer

The previous discussion has been related to the heat transfer associated with high speed flows with intact boundary layers in the absence of a pressure gradient. In many instances, at least locally, the boundary layer on a jet vane may be severely disrupted or even non-existent. A few references have been found that may be of use in predicting effects due to shock wave-boundary layer interactions, protuberances, and angle of attack. These are by no means the only complicating factors. Other details of the flow, which lie beyond the present coverage, include dissociation at high speeds and temperatures, surface ablation and other multi-phase multi-component effects, and boundary layer transition.

Shock_Wave-Boundary_Layer_Interactions.

When the thermodynamic equilibrium of a substance is perturbed, the effect of the disturbance is transmitted from molecule to molecule at the speed of sound. A new state of equilibrium is reached as the thermodynamic properties adjust to the requirements imposed by the disturbance. In an otherwise uniform flow, these requirements generally arise from changes in pressure that are typically caused by a deflection of the flow at a bounding surface or by the merging of two or more streams at different pressures.

In subsonic flows, propogation of a disturbance at the speed of sound permits a gradual achievement of the new state of equilibrium in regions both upstream and downstream of the disturbance. In supersonic flows, on the other hand, fluid

elements cannot receive the disturbance "message" until they have reached or passed the source of the perturbation. While the necessary adjustments occur smoothly in a subsonic flow, they occur suddenly in supersonic flows -- instantaneously in the theoretical limit of an inviscid non-conducting gas.

Disturbances demanding large pressure decreases in a supersonic flow are accomodated by sequences of small pressure decreases, and concomittant accelerations, that may be characterized as standing wave processes that are approximately isentropic. The region of adjustment is sometimes called an "expansion fan." On the other hand, when substantial pressure increases are required of a supersonic flow, partitioning of the necessary adjustment into a sequence of smaller pressure increases reveals that associated weak waves must coalesce into a single sudden event that is characterized by its name, the "shock wave".

Substantial dissipation occurs within a shock wave and in the limit the compression and deceleration processes are mathematically as well as physically discontinuous. For these reasons, the shock wave is not isentropic and its strength is directly related to the decrease in stagnation pressure of the gas passing through it. In a perfect gas, this decrease in stagnation pressure is easily related to the entropy increase across a shock.

Flows that are at supersonic speeds relative to their surroundings cannot stay that way indefinitely. As such flows are brought to rest, either intentionally or naturally, the

attendant decelerations must be accompanied by shock wave processes. Thus shock waves are always "waiting in the wings" in supersonic flows, and the task of the designer of devices involving supersonic flows (such as rocket nozzles) is to ensure that shock processes will not be overly detrimental to performance. In supersonic diffusers, for instance, careful design can lead to gradual slowing of the flow through a series of relatively weak shocks. Thus the cumulative pressure increase is attained with less loss than that associated with a single shock process.

The strength of shock waves (usually characterized by the concomittant pressure increase) depends upon the nature of their source. In the case of the deflection of an otherwise uniform supersonic flow by a rigid boundary, shock strengths increase with the deflection angle through the range of so-called "weak waves." These waves are oblique to the flow and, in inviscid flow, they emanate from the origin of relatively small deflections. Beyond a certain maximum deflection, depending on the upstream Mach number, shock waves detach from the deflection and become curved. In such "detached shock" cases, the flow downstream of the wave is extremely complicated. The curvature of the wave converts a one-dimensional upstream flow to a two-dimensional flow, with attendant nonuniformities in Mach number (including both subsonic and supersonic regions) and thermodynamic properties downstream of the shock.

While only weak shock processes are observed when oblique

shocks are attached to a deflecting boundary, the detached shock exhibits the entire range of strengths from the normal shock, through strong and weak oblique shocks, to waves of vanishing strength far from the disturbance. In cases where boundary conditions require flow adjustments to surrounding pressures, the strengths of the shocks occuring in supersonic flows will depend upon the extent of the necessary pressure adjustment. In such free-jet flows, compressive shock processes are usually interspersed with alternating expansions, as in rocket exhaust plumes adjusting to changing atmospheric pressures.

Finally, in this brief summary, it must be mentioned that shock and expansion waves seldom occur singly and without boundaries. When these waves arrive at rigid surfaces, they are changed in strength and, in most cases, reflected. Such occurences, involving shock intersections with boundaries, are the most common instances when inviscid approximations are likely to be unjustifiable. When waves intersect with one-another, there is a mutual modulation of their strengths. Flows downstream of such intersections are generally non-uniform and likewise subject to strong viscous effects.

Heat Transfer Aspects of Shock Wave-Boundary Layer Interactions. Boundary layers are happy in the presence of pressures that decrease in the direction of flow. When pressures increase, on the other hand, the pressure force supports the shear forces already active within the boundary layer. Such increasing pressures in the flow direction impose

thermocouple location. Figure 10 shows typical time histories of these temperatures, indicating the agreement of the temperature decay rates with expected exponential form. In the table below, results are shown for both sets of measurements. The experimental uncertainty for the two representative speeds shown was on the order of 2 Btu/hr-ft^2-R [39], indicating remarkably good agreement between the two methods -- virtually

Tunnel Speed ft/sec	Film Coefficient, h Btu/hr-ft^2-R	
	Thermocouple	IR System
200	27.5 - 33.2	27.7 - 33.5
306	41.0 - 42.7	39.9 - 41.8

identical within experimental uncertainty. The theoretical flat plate estimates for the film coefficient are 22 and 28 Btu/hr-ft^2-R for 200 and 306 ft/s, respectively, on the assumption of laminar flow. In both cases, this assumption is marginal, and the results indicate that turbulent or transitional flow probably existed at the location of the measurement. (This conclusion is further supported by the increased deviation at the higher speed.)

These tests served as a valuable learning experience. In hindsight, after encountering and working-out many experimental bugs, it seems probable that the planned initial testing in the supersonic wind tunnel was ill-conceived. In addition, these results validate the application of the transient method using IR thermography.

It is important to note that the measurements given above correspond to conditions at a particular point on the wedge -- the location of the thermocouple in this case. Further testing, with the cursor positioned at other points, would allow the development of a complete heat transfer map of the wedge surface. However, viewing of the thermal behavior of the entire wedge surface showed that the transient isotherm variation was essentially uniform over a large portion of the surface (with the exception of narrow bands in the three-dimensional boundary layer regions near the sidewalls). Thus it may be deduced that the values determined above, at the thermocouple location, are representative of conditions over a major portion of the wedge surface. The ability to make such observations, which might otherwise require the use of hundreds of simultaneous point measurements, is a significant virtue of the IR thermography method.

IR Thermography of a Jet Vane. The advantages of the IR thermography method are especially dramatic when it is used to view the thermal behavior of complicated configurations leading to flow effects that are well beyond exact analytical capabilities. Such is the case for the jet vane, and some preliminary testing was done in order to demonstrate the method.

The vane tested, shown mounted in the tunnel in Fig. 11, was constructed of copper-impregnated tungsten with an estimated volume-to-surface area ratio of about 1.5×10^{-4} ft. Thus, even though an exact value could not be calculated, the

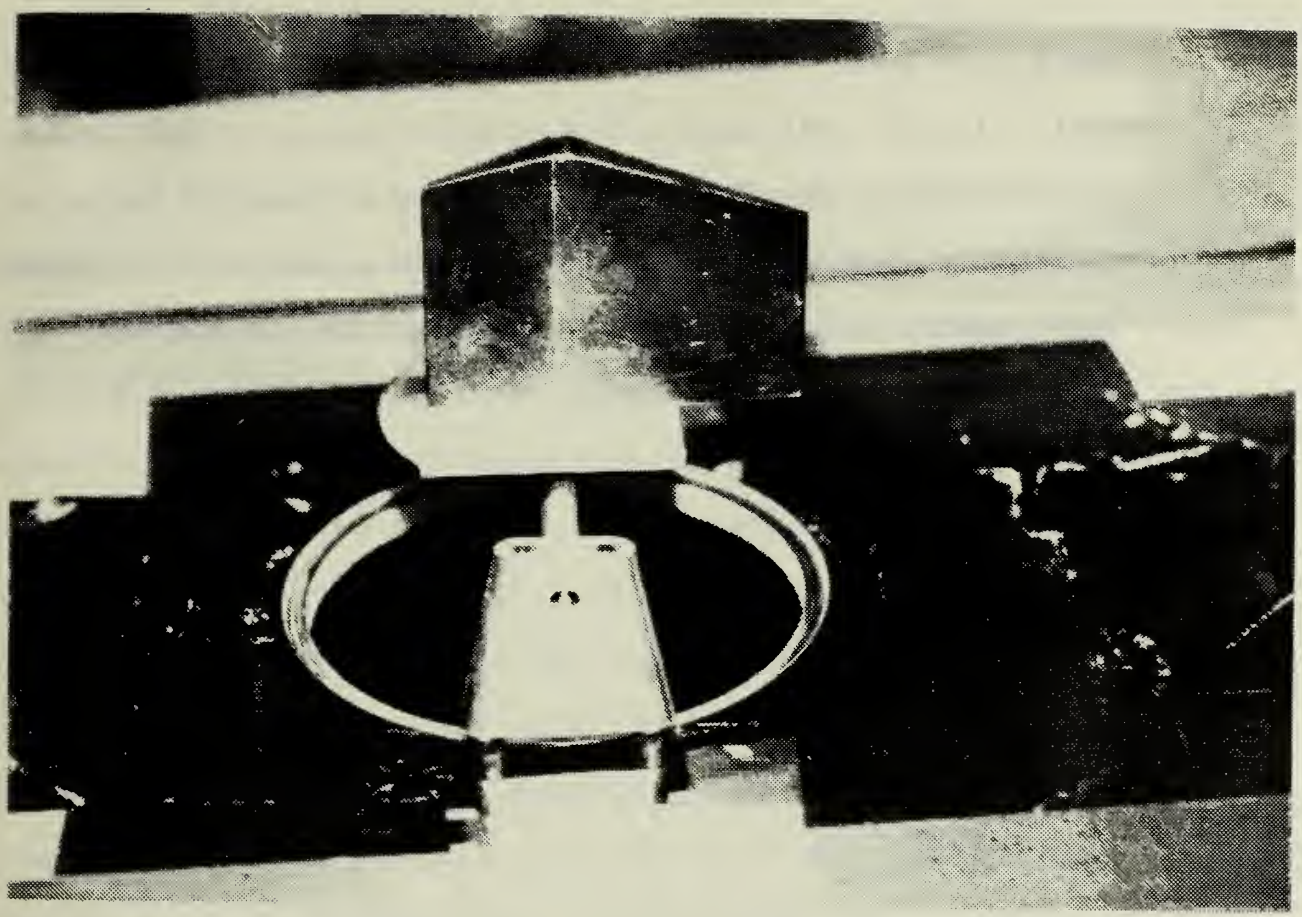

Figure 11. Photograph of jet vane model mounted in subsonic wind tunnel.

Biot number was well within the acceptable range. This was verified by the exponential temperature responses obtained in the tests. At 200 ft/s, the midspan heat transfer coefficient was measured to be 41.5 Btu/hr-ft^2-R, while a measurement at the vane/baseplate intersection indicated a 32% increase over this value.

By simply viewing the time history of the thermal response of the vane, it was possible to note the expected high heat transfer rates at the vane leading and trailing edges. Other areas of high heat transfer were noted just downstream of the ridge of the vane, particularly near the tip. In other

areas, the heat transfer appeared to be more-or-less uniform. Again, these tests proved the effectiveness of the method and indicated many possibilities for future research using this application of IR thermography. Further details regarding these tests, including color prints of the IR images, may be found in Spence [39].

CONCLUSION

Preliminary work has been carried out to develop the capability to model the heat transfer processes encountered by jet vane thrust vector control systems. Computational modeling has begun using the PHOENICS code. The initial configurations investigated were two-dimensional approximations to the NWC jet vane geometry. Two separate leading edge geometries have been investigated: a sharp wedge leading edge; and a rounded nose wedge. The results from these calculations indicate that the leading edge geometry has a very strong effect on the skin friction and surface heat transfer over the whole body. Work is continuing to investigate the effects of the leading edge geometry on the surface skin friction and heat transfer. It is believed that these effects stem from the complex shock wave boundary layer interaction near the leading edge. To carry out this investigation the latest version of the PHOENICS code (PHOENICS 84) is being procured. This version of PHOENICS will allow more precise modeling of various leading edge geometries as well as the modeling of three dimensional flow configurations.

In support of the computational modeling effort an experimental program has also been initiated. The purpose of this effort is to obtain confirmation of the computational results. The initial experimental work has been carried out in the subsonic wind tunnel and it is intended to continue the experimental work in the NPS supersonic wind tunnel.

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